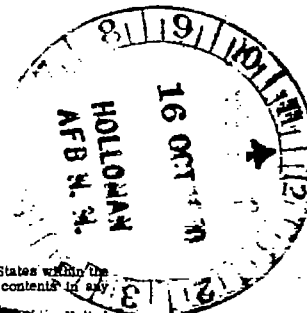NACA RM L50I13

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RESEARCH MEMORANDUM

INVESTIGATION OF THE PRESSURE-RATIO REQUIREMENTS OF

THE LANGLEY 11-INCH HYPERSONIC TUNNEL WITH

A VARIABLE-GEOMETRY DIFFUSER

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NATIONAL ADVISORY COMMITTEE
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SUMMARY

    Tests were conducted in the Langley 11-inch hypersonic tunnel at a
Mach number of 6.86 to determine the effectiveness of an adjustable
supersonic diffuser without boundary-layer control in reducing the pres-
sure ratios across the system required to maintain supersonic flow. The
results showed that the pressure ratio required to maintain supersonic
flow could be reduced to about one-third the pressure ratio required for
starting. This reduction quadrupled the running time of the tunnel.
The presence of models and model supports tended to increase the minimum
pressure ratio required to maintain flow. This effect, however, could
be minimized by selection of the optimum area ratio of the diffuser for
each model configuration.

INTRODUCTION

    Experience in the Langley 11-inch hypersonic tunnel at Mach numbers
of the order of 7 has shown that the minimum pressure ratio required to
maintain supersonic flow in the test section is considerably greater
than the stagnation pressure ratio across a normal shock if a simple
divergent diffuser is used. With a fixed-shape convergent-divergent
supersonic diffuser the possible reductions in required pressure ratio
are limited since only a relatively small decrease in channel area can
be allowed in order to avoid choking during the starting process. After
the flow in the tunnel has started and the normal shock has passed down-
stream of the diffuser throat, a much larger contraction can be allowed.
Ideally, the throat could be closed until a Mach number of unity existed
across the throat. Practically, however, shocks, viscous effects, and
associated unstable flow phenomena limit the minimum throat area to a
value considerably larger than the ideal; thus, the actual effectiveness
of the diffuser is well below that theoretically possible.

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Some method of starting the flow must be provided in order to take advantage of any decrease in the pressure ratio required to maintain flow. In the intermittently operated 11-inch hypersonic tunnel (reference 1) very large pressure ratios are available from a pressure-vacuum tank system for starting the flow. The variable-geometry diffuser is actuated immediately after starting in order to provide the lowest possible operating pressure ratio and hence the longest possible duration of test runs. These long runs are desirable in the intermittent type of wind tunnel in order to provide steady flow conditions and sufficient time for stabilization of manometers and other instruments. A tunnel of this type can be made continuous in operation through the addition of compressors designed for the relatively low pressure-ratio requirements for maintaining flow made possible through the action of the variable-area diffuser.

The purpose of the present tests was to determine the effectiveness of a variable-geometry diffuser designed for the 11-inch hypersonic tunnel as affected by diffuser area ratio, stagnation pressure, and the presence of test models and model supports. A two-dimensional nozzle of conventional design (reference 2) which produced reasonably uniform flow in the test section at a mean Mach number of 6.86 was used in these tests.

SYMBOLS

A_2 test-section area

A_3 area at upstream end of throat plates of diffuser

A_3' area at downstream end of throat plates of diffuser

p_s static pressure at surface of diffuser plates

p_4 stagnation pressure after diffuser

p_0 settling-chamber pressure

α angle of attack

θ plate angle relative to axis of diffuser

APPARATUS

The tests were conducted in the Langley 11-inch hypersonic tunnel (reference 1) equipped with the single-stage, two-dimensional nozzle of reference 2 and an adjustable-area diffuser located downstream of the section housing the model support strut. The nozzle test section was 10.51 inches in height by 9.95 inches in width and, consequently, the test-section area was 104.6 square inches. The nozzle, the strut section, and the variable-geometry diffuser are shown in figure 1.

In the ideal type of diffuser the air would be decelerated through a gradual compression; however, this would require accurately curved adjustable walls which are in general impractical in a supersonic diffuser. By decelerating the flow through two oblique shocks, as in this design, the compression may be accomplished with relatively small shock losses. Furthermore, the entrance plates were designed so that, in the full closed position at the design Mach number, the shock from the leading edge of the plate would meet the juncture of the opposite entrance and second minimum plates. This arrangement results in the flow being parallel to the axis as it enters the parallel channel between the second minimum plates, and also tends to minimize the effect of shock boundary-layer interaction since the shock is not reflected. The possibility exists, of course, that the flow in the diffuser might be quite different from the simplified model assumed for design purposes because of viscous effects.

Reference 3 indicated that a long constant area at the minimum section would result in a greater stability of the flow through the diffuser. A nearly constant-area section was therefore added after the contraction.

The angle of the diverging plates was kept small in order to obtain a high efficiency in the subsonic part of the diffuser.

The diffuser was necessarily adjustable to permit starting of the flow in the tunnel. Theoretically, from one-dimensional considerations, a diffuser throat about 63 percent of the test-section area would be sufficient at this Mach number; however, viscous effects and obstructions such as the model support strut upstream of the diffuser could appreciably alter the area required to allow starting. For this reason an arbitrary increase of 15 percent was made in the minimum diffuser area providing an area approximately 78 percent of the test-section area for the starting condition.

The size of the opening between the plates shown in figure 1 was maintained by the use of stops which restricted the movement of the pneumatically driven piston which powered the plates. One set of stops

determined only one diffuser opening. The minimum opening was limited by the lengths of the levers connecting the piston rod to the plates. The plates were hinged as shown in figure 1. The difference in over-all length at different area ratios was taken up by a sliding hinge at the downstream end of the diffuser plate. The plates with their actuating mechanism were housed in a pressure-tight case. The dead-air spaces between the case and the diffuser plates were vented to the stream at the rear of the diffuser plates. There was a gap between the side walls of the case and the plates of about 0.015 inch.

The dimensions of the parts controlling the diffuser openings were such that the central plates were not parallel except at the minimum area setting. (See fig. 2.) The maximum opening gave an area ratio between the test section and second minimum of 0.784, while the minimum opening gave an area ratio of 0.206. In figure 2, A_2 is the test-section area and A_3 and A_3' are, respectively, the areas at the upstream and downstream ends of the second-minimum plates. Also given in figure 2 are the angles of the entrance plates and the exit plates with respect to the axis of the diffuser.

METHODS

Pressures were obtained as close to the center line as possible along the lengths of the three plates composing the diffuser. Stiffening ribs prevented many of the orifices from being located on the center line, but none of the orifices were more than 1.3 inches off the center line of the plates.

The pressure instruments described in reference 1 were used for this survey. These instruments are bellows-type, 6-cell manometers, in which the deflection of the bellows is converted into the rotation of a small mirror which reflects a beam of light to a moving film thereby giving a time history of the pressure.

The pressures used to determine the pressure ratio required to maintain flow were the static pressure in the settling chamber and the pressure in the 2-foot-diameter pipe downstream from the diffuser as obtained from wall orifices at the moment of the breakdown of supersonic flow in the test section. The velocity in this pipe is low enough so that the static pressure there is negligibly different from the stagnation pressure.

Tests for the variation of the pressure ratio required with area ratio and the pressure distributions along the diffuser plates were made at a stagnation pressure of 30±1 atmospheres. Tests of the effect of model configurations in the test section were made at 26±1 atmospheres.

Additional tests were made at varying stagnation pressures in the range from about 25 down to 3 atmospheres. The stagnation temperature for the tests was $725\pm80^\circ$ Fahrenheit, a value high enough to maintain the static temperature in the test section well above the normal static liquefaction point of air at the test-section pressure. A regulating valve at the reservoir tank maintained the stagnation pressure in the tunnel constant throughout the running time.

RESULTS AND DISCUSSION

Pressure ratio required to maintain flow.- The results of figure 3(a) show the very substantial effectiveness of the diffuser. The theoretical curve on this plot was obtained by assuming one-dimensional isentropic flow between the test section and the diffuser throat and a normal shock in the diffuser throat. The experimentally determined pressure ratio required to maintain flow is seen to be from approximately 60 to about 115 percent higher than that given by the theoretical line because of the presence of viscous and shock effects. Measurements of the boundary layer in the test section given in reference 2 indicate that the displacement thickness of the boundary layer is about 3/4 of an inch. Of course the boundary layer, some 45 inches downstream, at the entrance of the diffuser, would be considerably thicker. The theoretical pressure ratio required without a diffuser $\left(\dfrac{A_3}{A_2} = 1.0\right.$ on fig. 3(a)$\Big)$ corresponds to the total pressure ratio across a normal shock at the test-section Mach number. The pressure ratio required for the smallest throat area tested is less than half of the theoretical normal shock pressure ratio.

With the maximum throat opening, a pressure ratio of 79 was required to maintain supersonic flow in the unobstructed nozzle or about $1\frac{1}{3}$ of the normal shock value. From an extrapolation of the experimental curve in figure 3(a), to $\dfrac{A_3}{A_2} = 1.0$, a pressure ratio of about 97 is seen to be required to maintain flow without any diffuser. This value is about $3\frac{1}{3}$ times as great as the pressure ratio of about 29 required at the smallest throat area tested. Further reductions in area might be beneficial for some test configurations.

Figure 3(b) illustrates the effect of an obstruction ahead of the diffuser on the pressure ratio required to maintain flow. The obstruction in this case was the diamond-shape model support strut upon which was mounted a 10° included angle cone located in the center of the test section. The ratio of the unobstructed area at the strut's maximum thickness to the test-section area was 0.779. The effect of the strut, as shown by a comparison of figures 3(a) and 3(b) was to make the

diffuser more effective at its larger openings and somewhat less effective at its smaller openings. The optimum area ratio in this condition is about 0.304 with a minimum pressure ratio of about 43 required to maintain flow. When the area ratio between the second minimum and the test section was reduced below 0.304 large increases in the required pressure ratio resulted. At an area ratio between 0.206 and 0.215 a complete breakdown of supersonic flow occurred immediately upon closing the diffuser. The mechanism of this phenomenon will be discussed later in more detail. The number of tails on some of the data points in figure 3 indicate the number of additional test runs the results of which were in agreement.

The effect of test models.- Figure 4 illustrates the effect that two wing models (4 by 4 in. in plan form) had upon the required pressure ratio. The effects are noticeable, but small, until the angle of attack is sufficiently high so that the wake and shock losses of the wing cause choking in the diffuser throat. This condition manifested itself as a sudden large decrease in the length of test run as the angle of attack is increased. As shown in figure 4, this effect can be delayed to a higher angle of attack by enlarging the area of the second minimum. From the results of figures 3 and 4, it is seen that no single optimum area ratio exists for all test configurations. --

Effect of pressure ratio required on the duration of test run.- Figure 5 presents a plot of the duration of test run in seconds as a function of the pressure ratio required to maintain flow. This plot makes it evident that for the condition of figure 3(a) the running time is increased by a factor of 4 in changing the configuration from the case where there is no reduction in the diffuser-throat area from the test-section area to the smallest diffuser-throat area tested. With the model support strut and the 10° cone ahead of the diffuser the optimum pressure ratio required was about 43. For this configuration the running time was increased by a factor of about 3.

In figure 5, no attempt was made to correlate the initial conditions of temperature, pressure, and pressure ratio across the system at the start of the run with the parameters of the graph. Thus what has been faired as one curve in the figure is really a family of curves. In general, this variation in initial conditions was not great and its effect on the length of run was small.

Pressures in the diffuser.- Figure 6 presents the pressures as measured along the approximate center line of the plates forming the diffuser with the condition of figure 3(a), that is, without the model support. These pressures are given in the form of the plate surface pressure divided by the settling-chamber pressure. In this figure, the pressure curves that are referred to as "break" mean the pressures taken the instant before the breakdown of supersonic flow in the test section

of the nozzle. The pressure ratios of these break curves correspond to
the values given in figure 3(a). The other curves represent the pres-
sure distributions at various times during the run. By referring to
figure 5, the pressure ratios chosen $\left(\dfrac{p_0}{p_4} = 105, 70, \text{and } 50\right)$ are seen
to correspond to times of approximately 20, 40, and 60 seconds, respec-
tively, after the start of a run. In figure 6, the gradual progress of
the shock upstream along the diffuser may be followed. The pressure
gradient near the shock becomes steeper as the shock moves upstream.
When the shock reaches the upstream portion of the throat and the pres-
sure ratio across the system is too low to keep the shock in the throat
an abrupt breakdown of the supersonic flow occurs. Of course, what has
been referred to as "the shock" is probably a complex system of shocks
since the pressure rise in most cases is gradual even for the curve
representing the conditions immediately preceding the break.

Though not readily seen due to the scale of figure 6, the pressures
over the entrance plate of the diffuser are from $1\frac{1}{2}$ times the test-
section pressure at the larger throat areas to $3\frac{1}{2}$ times the test-section
pressure at the smaller throat areas. These pressure rises are close to
those predicted by the oblique shock theory where the flow-deviation
angle is assumed equal to the incidence angle of the plates.

Figure 7 presents the diffuser-wall pressure distributions obtained
with the diamond-shape model support strut in place and the 10° included
angle cone installed in the tunnel in the same configuration as for fig-
ure 3(b). The curve for $\dfrac{A_3}{A_2} = 0.238$ of figure 7 represents a condition in
which the area of the throat is below the optimum, that is on the portion
of the experimental curve in figure 3(b) which has a negative slope. In
figure 7, if the pressures at the area ratio of 0.238 be compared
with 0.358, a striking difference is found. At the area ratio of 0.358
the shock in the diffuser shows a gradual progress upstream, as the
pressure ratio decreases, with an increasing gradient of pressure as
was noted in figure 6. At the area ratio 0.238, however, the shock does
not progress upstream into the throat, but appears to move only a short
way along the diffuser exit plate when the diffuser chokes and thereby
causes complete breakdown of the flow. The direct cause of this choking
is probably leakage through the gap between the side walls and the plates
from the dead-air chambers where the mechanism is housed. The source of
high-pressure air is the air downstream of the normal shock on the
diffuser plate. This leakage of course is present at all area ratios
and all configurations to a varying extent. But the amount of leakage
that can be tolerated without choking is considerably decreased by the
presence of the wake and shock losses from the support strut.

Except for area ratios below the optimum, the pressure changes shown in figure 7 are much the same as those in figure 6, where the nozzle and diffuser were unobstructed. However, it will be noticed that at the larger area ratios, there are noticeable effects of disturbances being carried upstream and affecting the pressures along the entrance plates just before the breakdown of supersonic flow. These disturbances, however, have no detectable effect upon the pressures measured in the test section. With the model support strut in place the pressure on the entrance plate is from $2\frac{1}{2}$ to 4 times the test-section pressure depending on the diffuser opening.

Effect of varying the stagnation pressure on the pressure ratio required to maintain flow.- Figure 8 presents the variation of the pressure ratio required to maintain flow together with the variation of the average Mach number in the central useable portion of the test section as a function of the stagnation pressure of the nozzle. (See reference 2 for a more complete study of the pressure effect.) The Mach number remains essentially constant as the stagnation pressure is reduced from 25 atmospheres to a stagnation pressure of about 14 atmospheres. Below this pressure there commences a gradual decrease in the test-section Mach number which continues to the lowest stagnation pressure tested, about 3 atmospheres. The pressure ratio required to maintain flow also appears to follow this trend, being essentially constant as the pressure is reduced to approximately 14 atmospheres, then decreasing as the stagnation pressure is reduced further. Thus, it appears that the stagnation pressure affects the pressure ratio required to maintain flow mainly through its effect upon Mach number.

CONCLUDING REMARKS

The use of a variable-area supersonic diffuser in the Langley 11-inch hypersonic tunnel made it possible in some cases to maintain flow in the nozzle with stagnation pressure ratios across the system as low as one-third of that required without a second minimum with a consequent quadrupling of the running time. In a continuous running tunnel started by a pressure-vacuum tank system as employed in the 11-inch hypersonic tunnel, the use of this diffuser would appreciably reduce the installed power and the size and number of stages of the drive compressors.

The presence of test models and their supports tended to increase the pressure-ratio requirements. This effect, however, can be minimized by selecting the optimum area ratio of the diffuser for each test configuration.

No attempt was made in the present series of tests to improve the performance of the diffuser by reducing the flow leakage known to exist around the edges of the adjustable plates. Furthermore, the minimum area of the diffuser was limited by the mechanical linkages used and further reductions in area might be beneficial for some test configurations.

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2. McLellan, Charles H., Williams, Thomas W., and Beckwith, Ivan E.: Investigation of the Flow through A Single-Stage Two-Dimensional Nozzle in the Langley 11-Inch Hypersonic Tunnel. NACA TN   , 1950.

3. Kantrowitz, Arthur: The Formation and Stability of Normal Shock Waves In Channel Flows. NACA TN 1225, 1947.
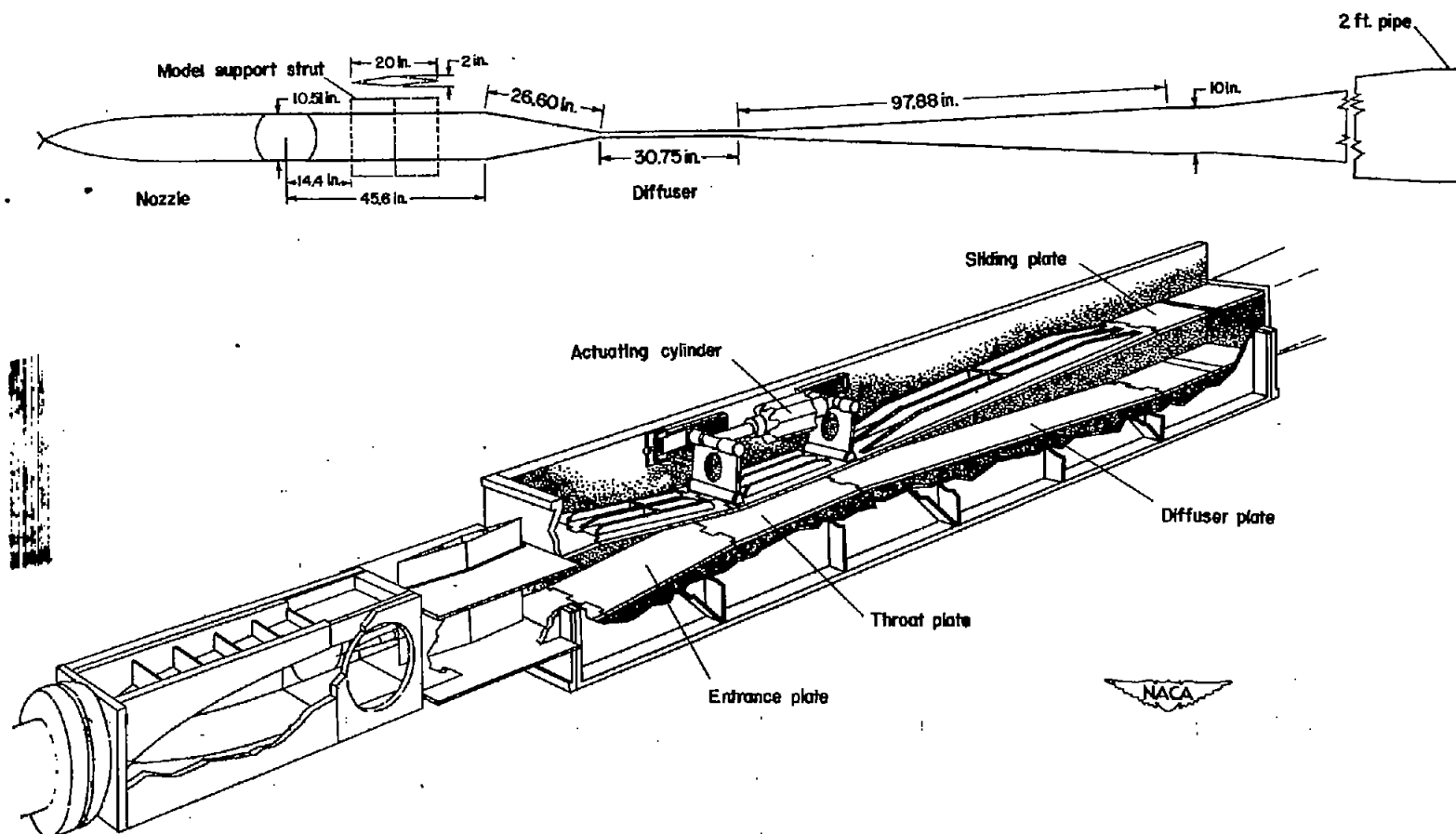
Figure 1.- Sketch of diffuser and nozzle with model support strut in place.
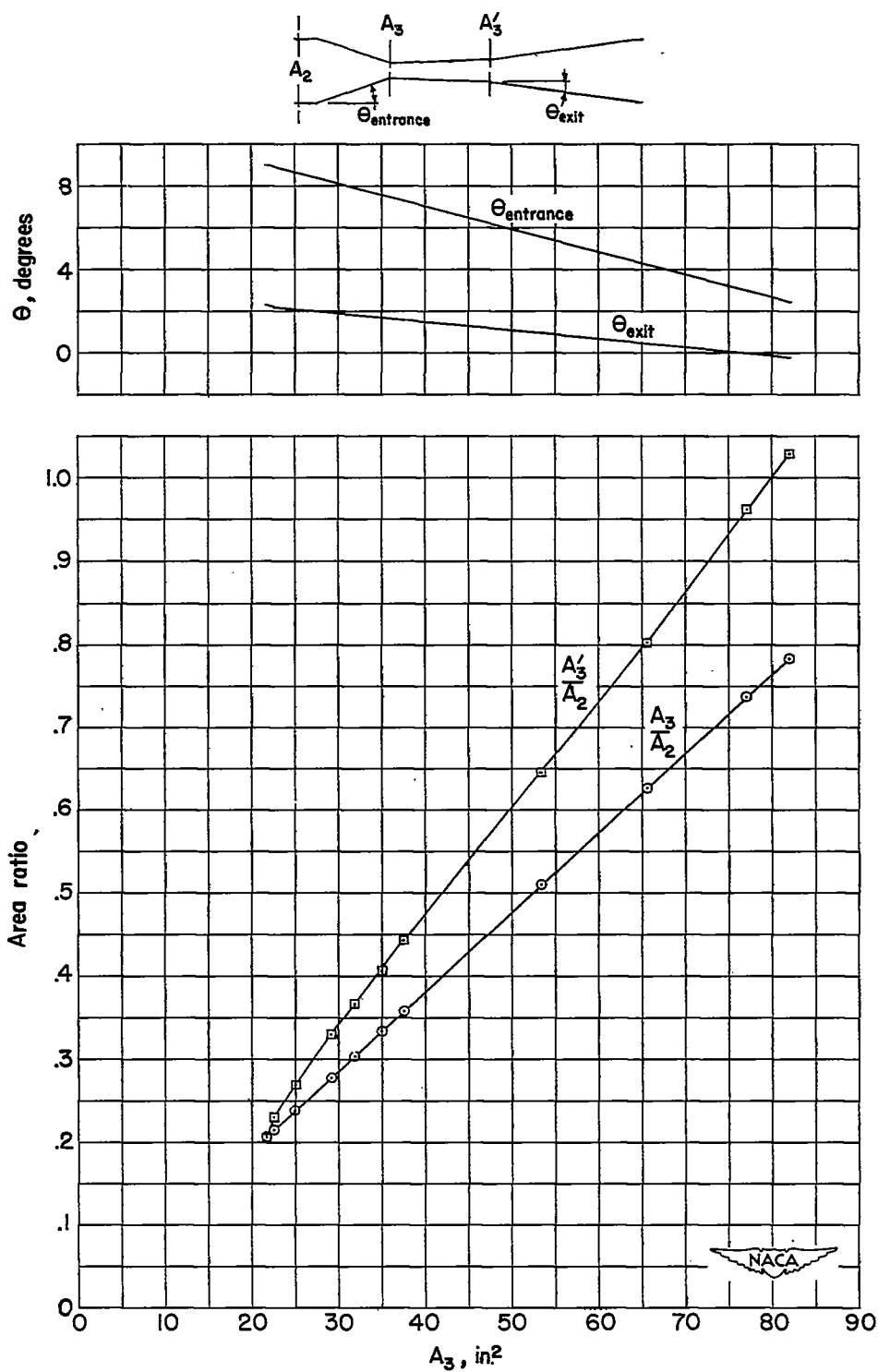
Figure 2.- The areas in the diffuser determined by the diffuser throat
plates and the angle of incidence of the entrance and exit plates.
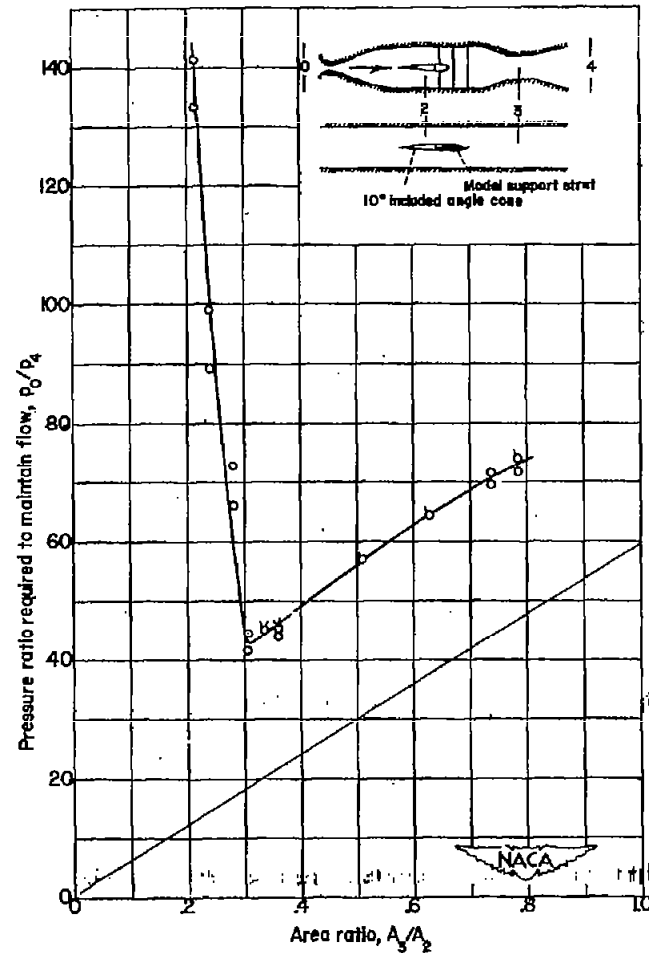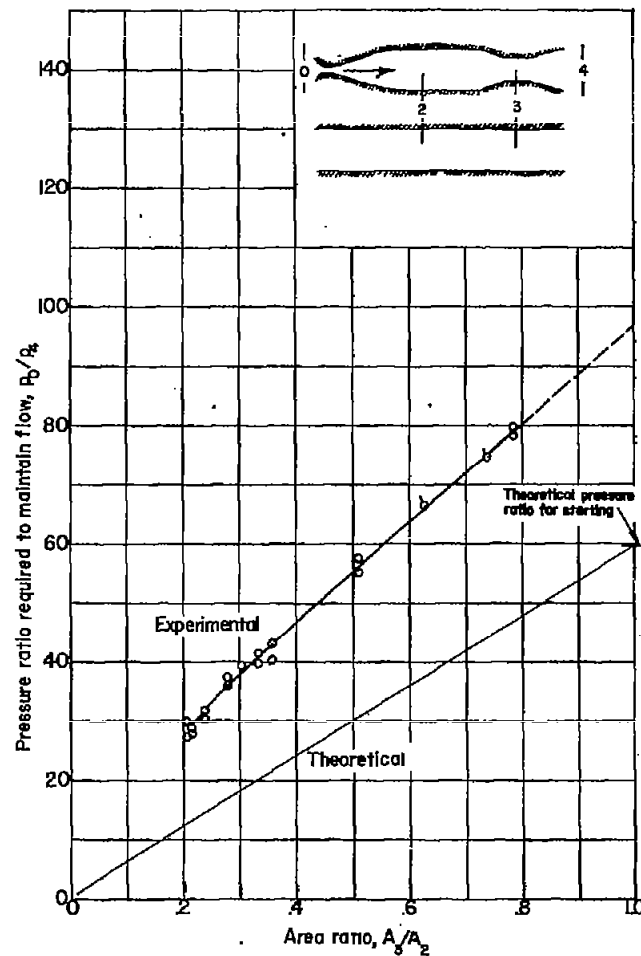
(a) Unobstructed nozzle and diffuser.   (b) Nozzle and diffuser with model
support strut.

Figure 3.- The pressure ratio across the system required to maintain super-
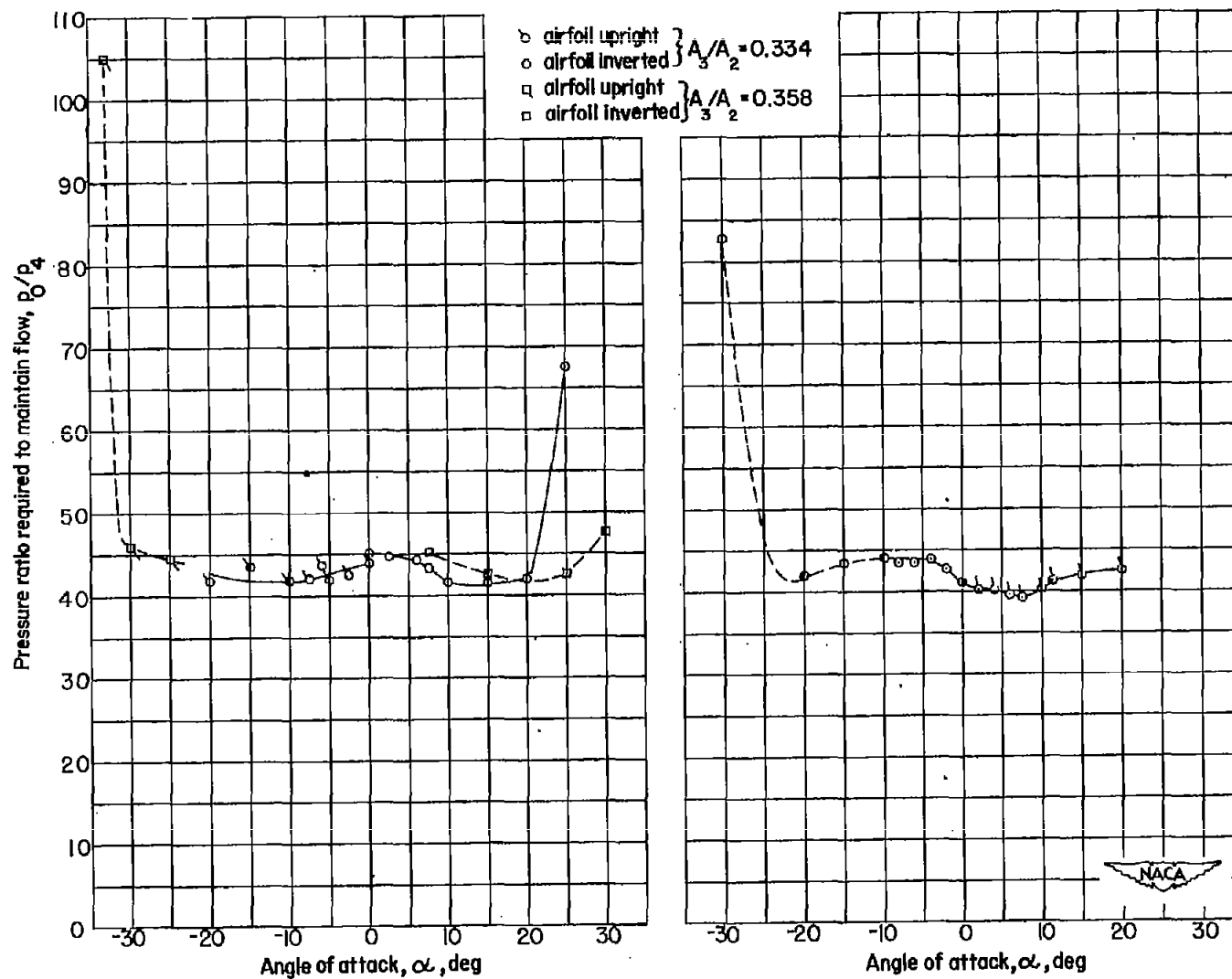sonic flow as a function of the area ratio between the diffuser throat
and the test section.

Figure 4.- The effect of two airfoils at various angles of attack on the pressure ratio required to maintain flow.
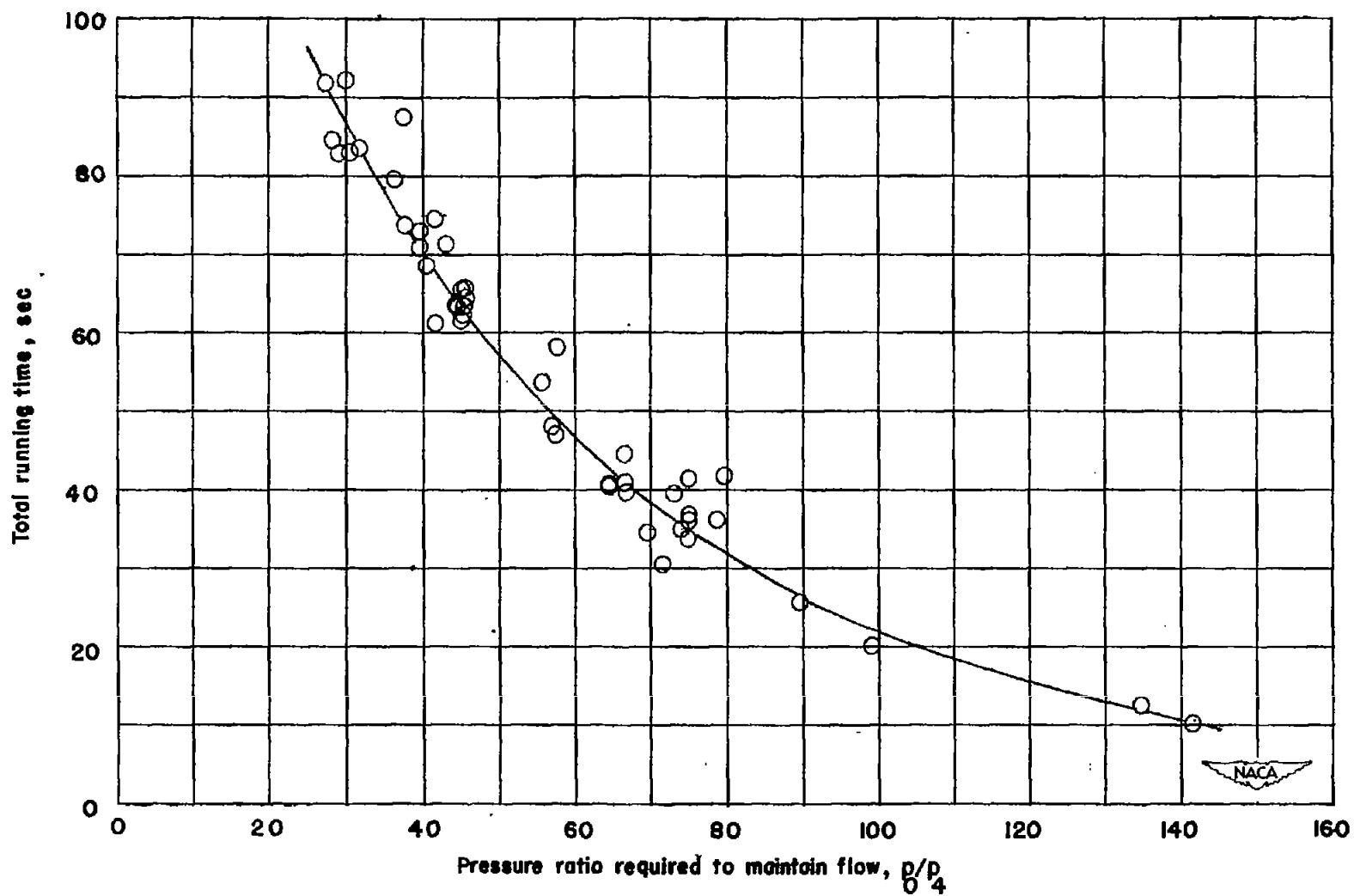
Figure 5.- The duration of running time as a function of the pressure
ratio required to maintain flow.
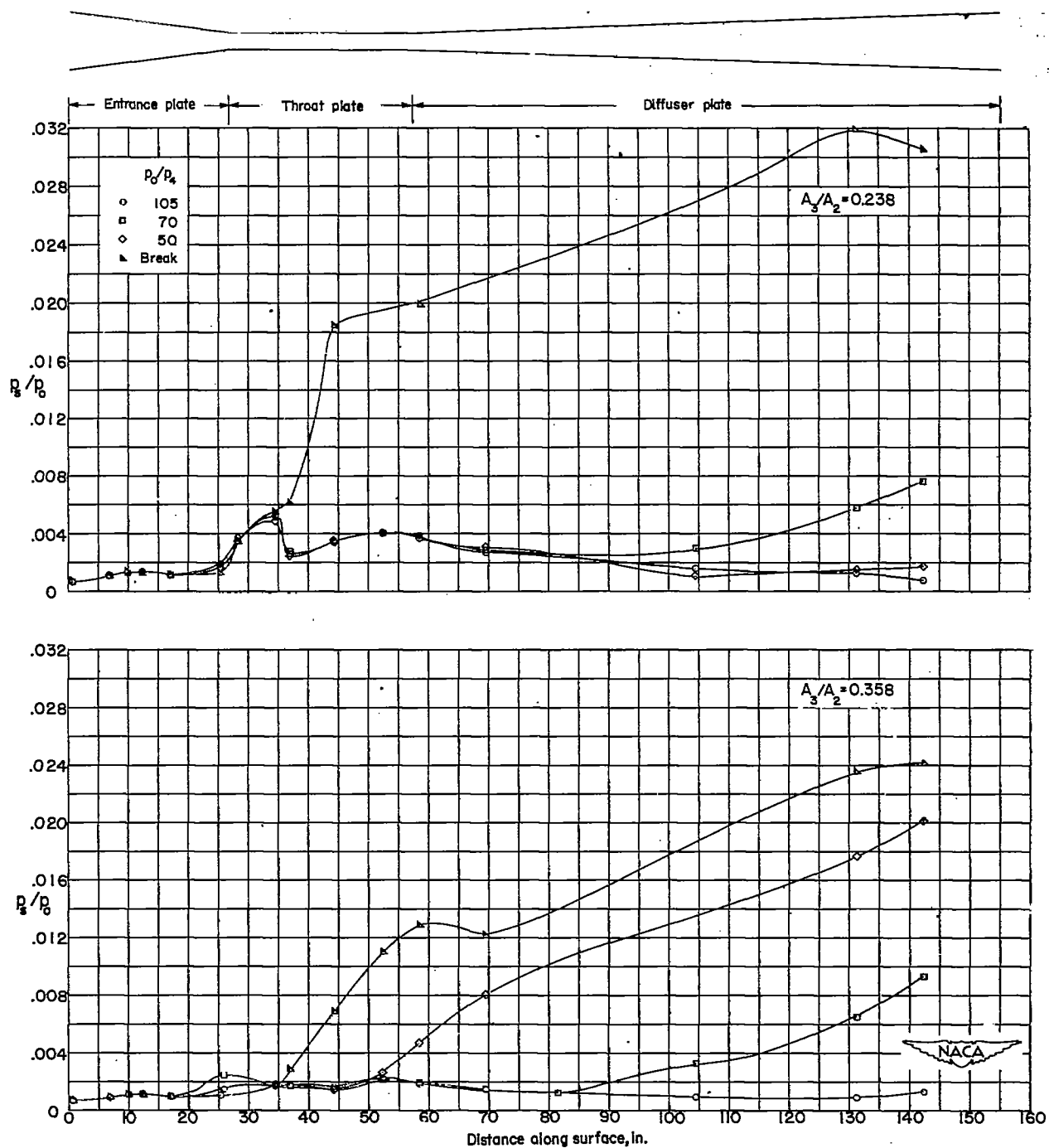
Figure 6.- The pressures along the plates of the diffuser with no
obstruction between the nozzle and diffuser for various pressure
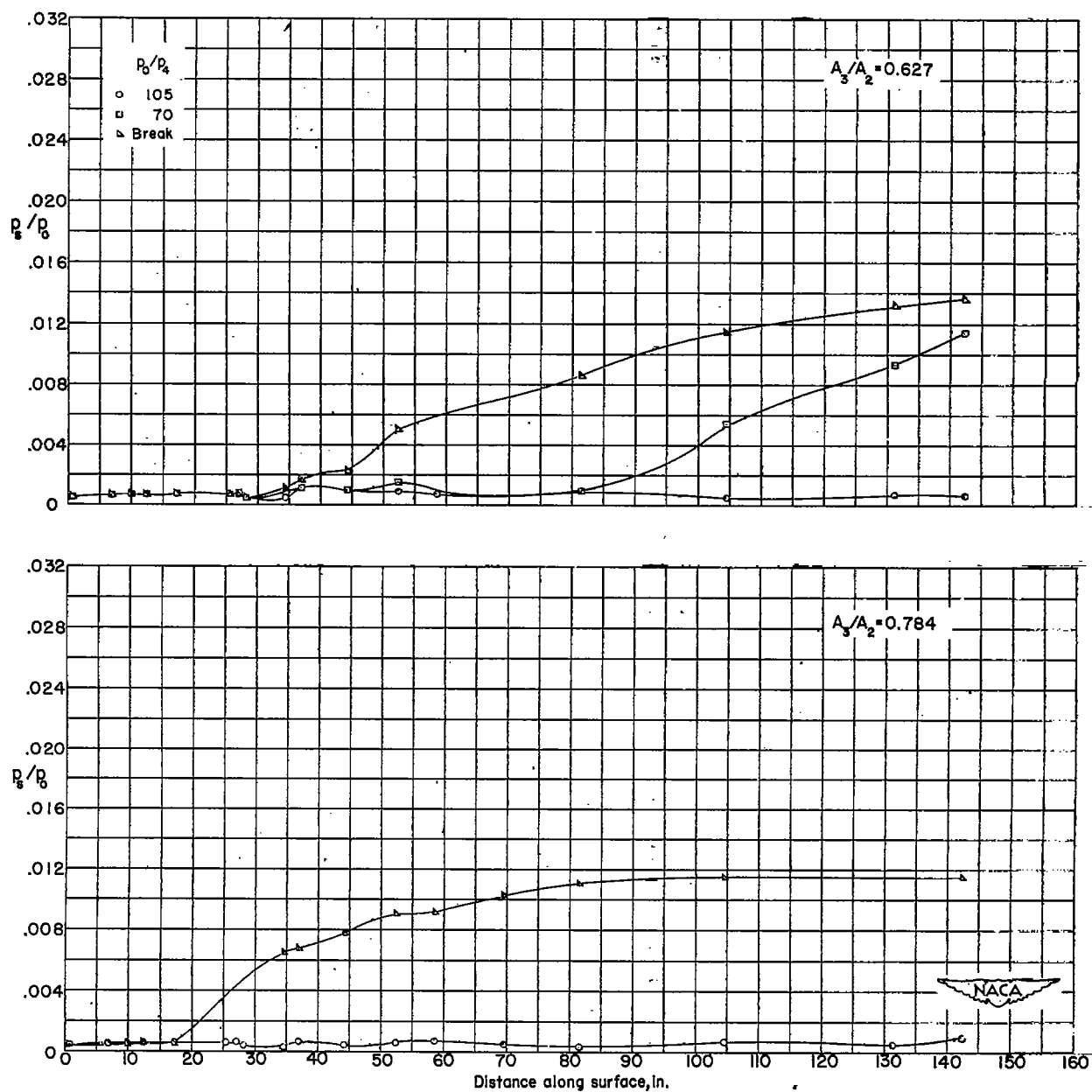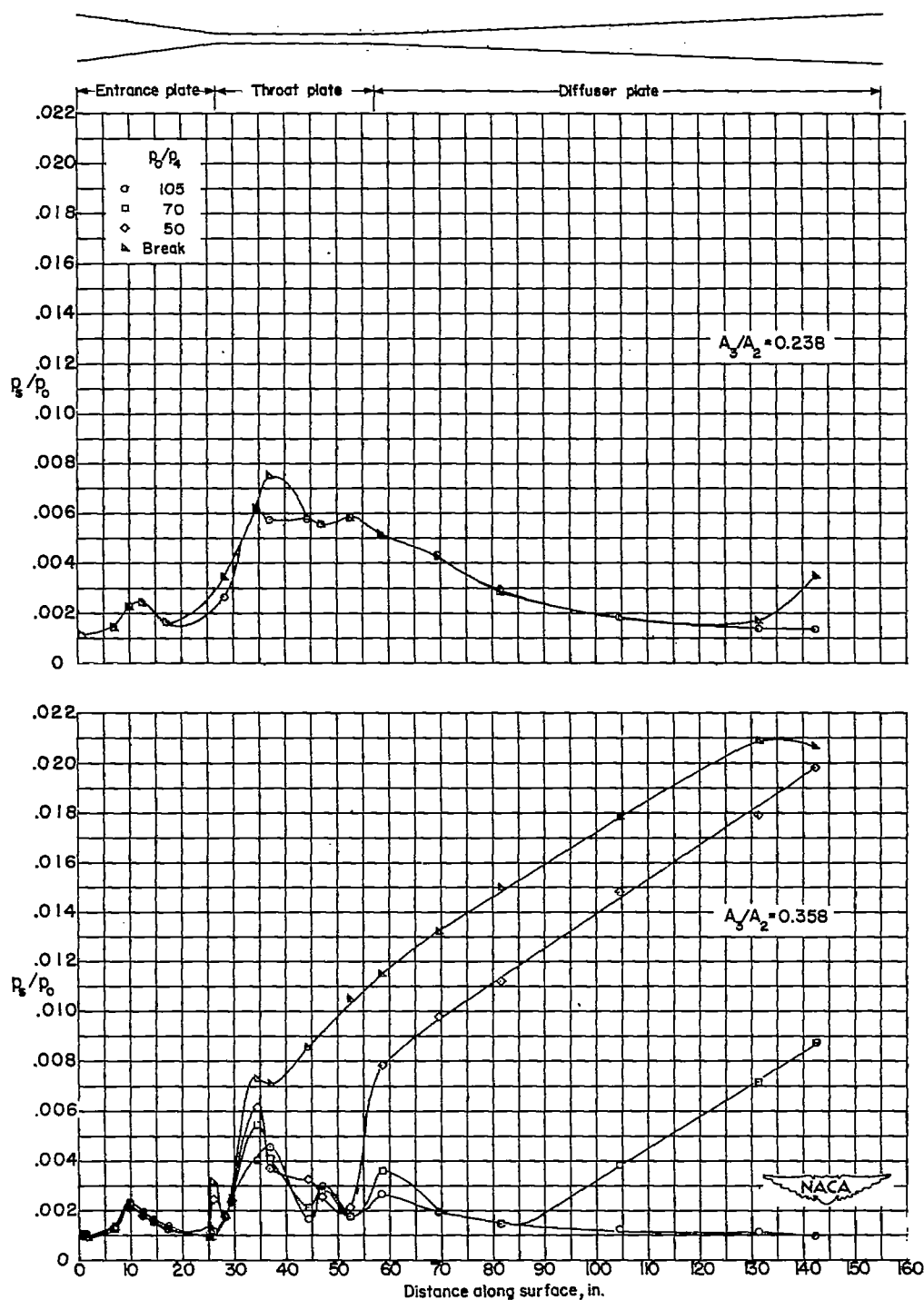ratios across the system at different area ratios.

Figure 6.- Concluded.

Figure 7.- The pressures along the plates of the diffuser with the model
support strut and a 10°-included-angle cone in place for various
pressure ratios across the system at different area ratios.

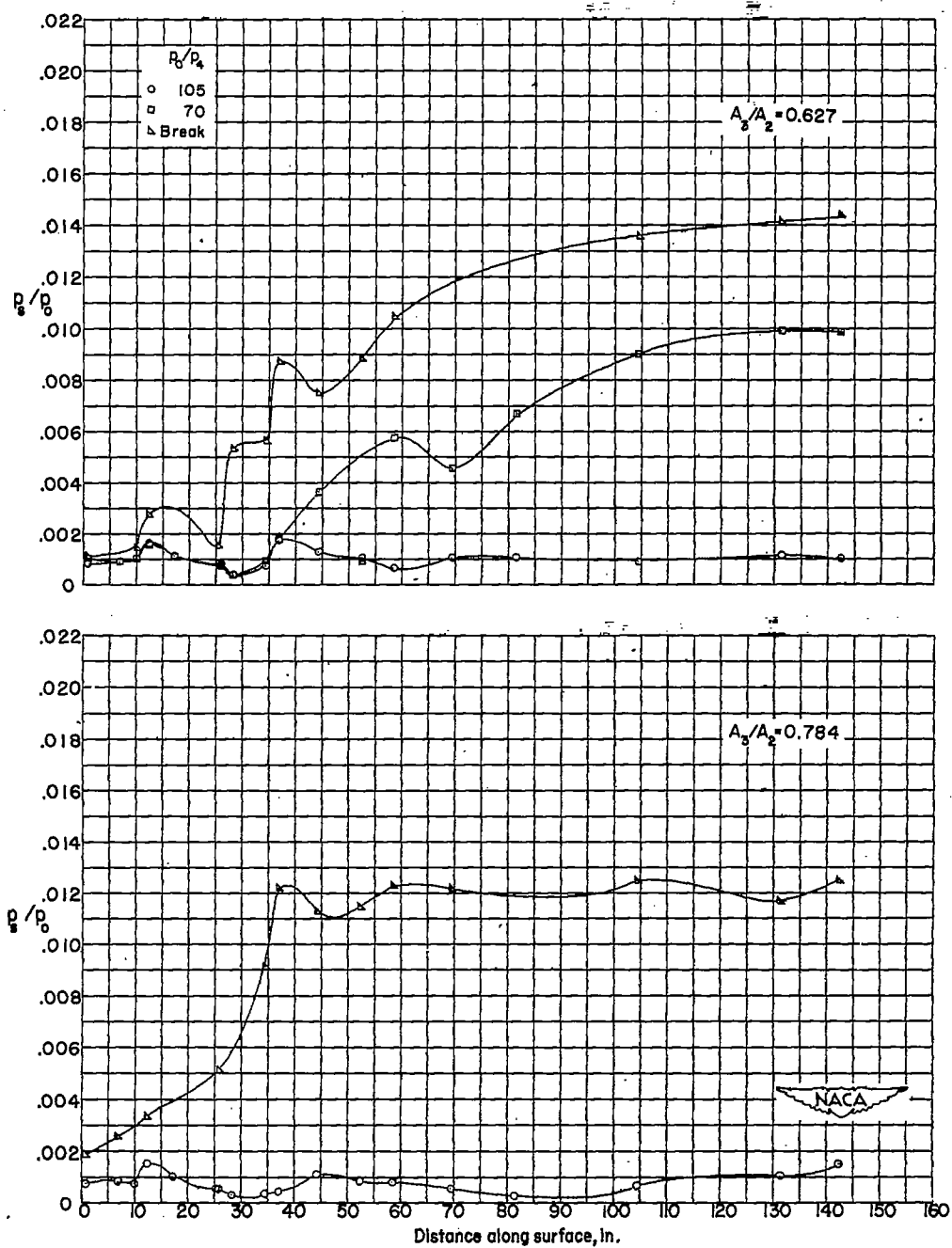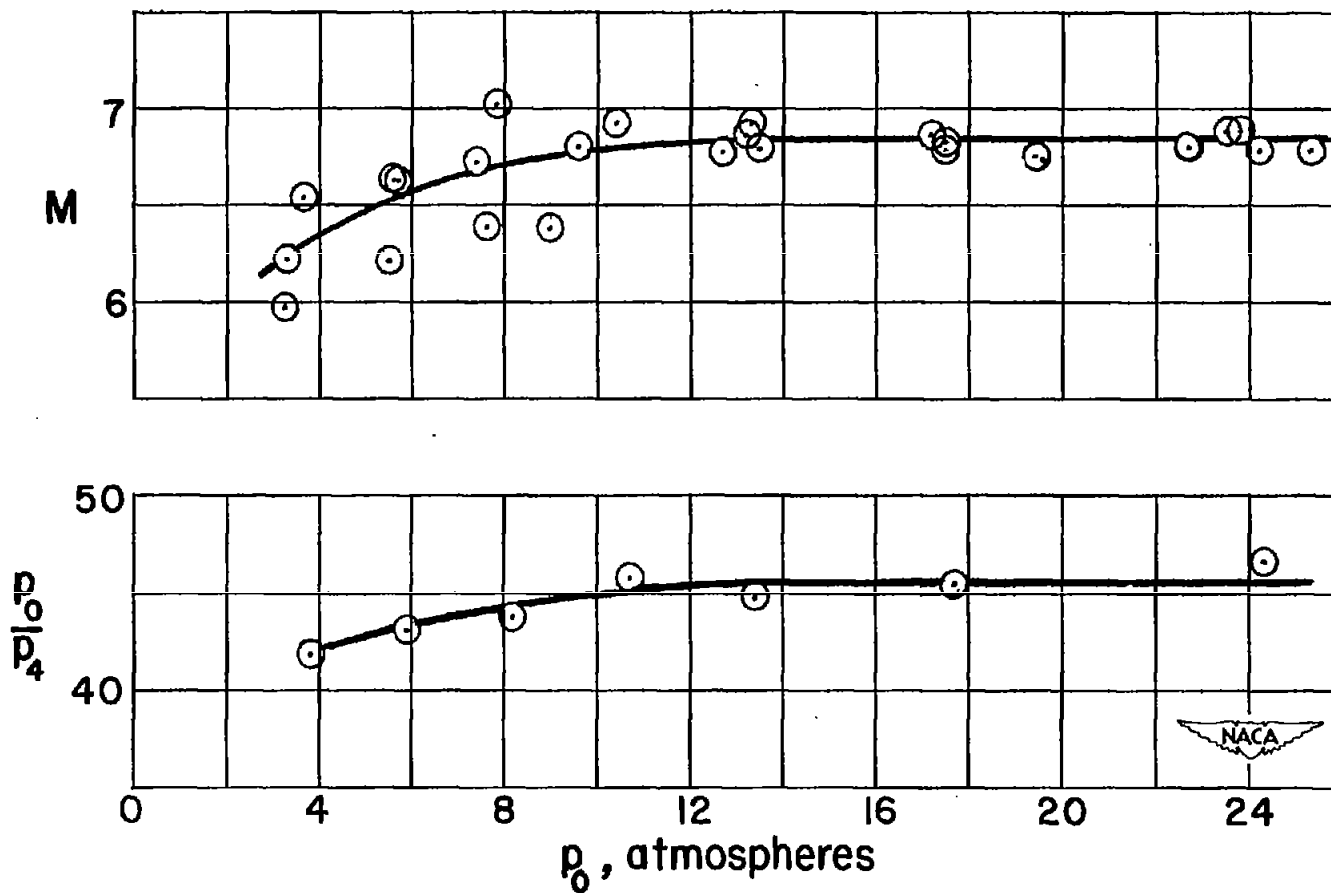Figure 7.- Concluded.

Figure 8.- The effect of varying the stagnation pressure on the test section Mach number and the pressure ratio required to maintain flow. $\dfrac{A_3}{A_2} = 0.358$.